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RESEARCH MEMORANDUM

PERFORMANCE OF J33 TURBOJET ENGINE WITH SHAFT-POWER EXTRACTION

I - OVER-ALL ENGINE PERFORMANCE

By J. C. Nettles and J. R. Easterly

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RESEARCH MEMORANDUM

PERFORMANCE OF J33 TURBOJET ENGINE WITH SHAFT-POWER EXTRACTION

I - OVER-ALL ENGINE PERFORMANCE

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SUMMARY

The performance of a centrifugal-compressor turbojet engine operated at constant speed and variable turbine-inlet temperature by means of a dynamometer is presented. Data from operation as a conventional jet engine are presented for comparison. The amount of shaft horsepower that could be extracted from the engine with a tail-pipe-outlet area 22 percent larger than that of the standard engine and a turbine-inlet temperature of 2000° R was 1650 horsepower per square foot of turbine-nozzle area at a compressor tip speed of 1293 feet per second and decreased slightly at higher tip speeds. The ratio of shaft horsepower to total available horsepower decreased from 0.512 at a compressor tip speed of 1023 feet per second to 0.313 at a compressor tip speed of 1438 feet per second. If a propeller is added to the engine, an increase of better than two and one-half times the static thrust is possible at a compressor tip speed of 1438 feet per second. The thrust specific fuel consumption with a propeller would be 0.58 pound fuel per pound of thrust at the compressor tip speed of 1438 feet per second.

INTRODUCTION

Investigations of the performance of turbojet engines with fixed tail-pipe areas in which the turbine extracts just sufficient power to drive the compressor and accessories are generally restricted to a fixed relation between rotor speed and turbine-inlet temperature. If a power absorbing means is added to the turbine shaft, the

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control of the engine speed is independent of turbine-inlet temperature from some minimum value to the maximum that the turbine stress and material limits will permit.

An investigation was conducted at the NACA Cleveland laboratory at the request of the Bureau of Aeronautics, Department of the Navy, in order to determine the shaft horsepower that was available from a jet engine having a centrifugal-flow compressor, 14 cylindrical type combustion chambers, and a single-stage turbine, as well as to determine the trends of the performance parameters of the engine components. The excess shaft horsepower that is available from the engine is applicable to a number of uses such as the driving of a propeller to increase take-off thrust, the removal of boundary-layer air, or the driving of accessory pumps for rocket motors. In this investigation, the engine was considered as a turbine-propeller engine.

A jet engine was converted for shaft-horsepower extraction by coupling a dynamometer to the engine shaft thus permitting engine performance to be studied over a wide range of turbine-inlet temperatures for any selected engine speed. The standard jet nozzle was removed in order to increase the pressure drop across the turbine.

SYMBOLS

The following symbols are used in this report:

c_p specific heat of fluid at constant pressure, Btu per pound per $^{\circ}R$

F thrust, pounds

g acceleration of gravity, 32.2 feet per second per second

h_f lower heating value of fuel, Btu per pound

hp horsepower

Δh enthalpy change, Btu per pound

J mechanical equivalent of heat, 778 foot-pounds per Btu

N engine speed, rpm

P total pressure, pounds per square foot shp shaft horsepower

T total temperature, °R

U tip speed, feet per second

W weight flow, pounds per second

γ ratio of specific heats

δ ratio of pressure to standard NACA sea-level pressure

η efficiency

θ ratio of temperature to standard NACA sea-level temperature

φ ratio of shaft horsepower to total available power

Subscripts:

1 compressor inlet

2 compressor outlet

3 turbine inlet

4 turbine outlet

A ideal work available by expansion to ambient pressure

a air

b burner

c compressor

e thermal

f fuel

g gas (air plus fuel)

j jet

s shaft

t turbine

tot total

PERFORMANCE PARAMETERS AND ANALYSIS OF DATA

In the cycle under investigation, air is compressed in the compressor, heat is added at constant pressure in the burner, and the gas is expanded through a turbine and jet nozzle. The horsepower available for driving a shaft is the difference between the turbine horsepower and compressor horsepower. The shaft horsepower depends on certain design criterions, which determine the manner in which the heat available for doing work is divided between the turbine and the jet.

The thermodynamic relations for the work of a polytropic process lead to the following expressions, which are useful in determining the horsepower available from a combination of compressor, constant-pressure burner, and turbine.

Compressor horsepower. - The compressor horsepower is

$$hp_c = \frac{W_a J c_p T_1}{550\ \eta_c}\left[\left(\frac{P_2}{P_1}\right)^{\frac{\gamma-1}{\gamma}} - 1\right] \tag{1}$$

Turbine horsepower. - The turbine horsepower is

$$hp_t = \frac{W_g J c_p T_3}{550}\ \eta_t\left[1 - \left(\frac{P_4}{P_3}\right)^{\frac{\gamma-1}{\gamma}}\right] \tag{2}$$

Burner temperature rise. - The temperature rise in a burner with constant-pressure combustion may be written as

$$\Delta T_b = \frac{W_f h_f}{W_a c_p}\ \eta_b \tag{3}$$

Equivalent shaft plus jet horsepower. - Although the horsepower derived from a jet at static conditions is zero, an expression for jet thrust and shaft horsepower can be written in terms of the enthalpy change of the working fluid. This expression credits the engine with the total thermodynamic power available and can be arranged as follows:

$$F = \frac{W_g}{g} \sqrt{2gJ\Delta h} \tag{4}$$

$$\Delta h = \frac{F^2 g}{2 \cdot W_g^{\,2} J} \tag{5}$$

$$hp = \frac{W_g J \Delta h}{550} \tag{6}$$

$$\Delta h = \frac{550\ hp}{W_g J} \tag{7}$$

Substituting equation (5) into equation (6) gives

$$hp_j = \frac{F^2 g}{1100\ W_g} \tag{8}$$

which is the horsepower that would have been developed had the enthalpy drop of the jet been utilized in a turbine.

The total equivalent shaft horsepower of the engine may therefore be written

$$hp_{tot} = shp + \frac{F_j^{\,2} g}{1100\ W_g} \tag{9}$$

This equation does not imply that the enthalpy drop of the jet is actually available for a turbine because the turbine efficiency is of a lower order than the nozzle efficiency; a higher pressure ratio would therefore be required to develop a given enthalpy drop. The relation expressed in equation (9) does give a rational basis to account for both thrust and horsepower in a single term and is essentially independent of whether the engine is entirely jet, entirely propeller, or a combination of the two.

Thermal efficiency. - Substituting equations (5) and (6) into the expression for thermal efficiency of the engine gives

$$\eta_e = \frac{550\ hp_{tot}}{JW_f h_f} \tag{10}$$

$$\eta_e = \frac{1}{JW_f h_f}\left(550\ shp + \frac{F_j^{\,2}\ g}{2W_g}\right) \tag{11}$$

<u>Ratio of shaft horsepower to total horsepower.</u> - In a gas-turbine engine, the ratio of external shaft work to the total thermodynamic work available from the engine is important. This statement is particularly true for turbine-propeller engines because the optimum ratio of shaft work to total thermodynamic work available varies with flight speed. An expression for the ratio of shaft work to total available work in terms of enthalpy change is:

$$\varphi = \frac{\Delta h_t - \Delta h_c}{\eta_t\ \Delta h_A - \Delta h_c} \tag{12}$$

where Δh_A is the maximum ideal energy available by adiabatic expansion based on turbine-inlet total conditions and ambient static conditions. This expression for φ is readily adaptable to cycle-analysis procedures but is not easily obtained from experimental data because of the difficulty of determining Δh_t and η_t. An approximate expression for φ, which is readily obtained from experimental data, is

$$\varphi = \frac{\Delta h_s}{\Delta h_s + \Delta h_j} \tag{13}$$

$$= \frac{1}{1 + \dfrac{F_j^{\,2}\ g}{W_g\ shp\ 1100}} \tag{14}$$

This value of φ may vary slightly for a given temperature and pressure ratio because the expansion process for determining the available energy involves both the efficiency of the turbine and the efficiency of the jet nozzle. The advantage of using the approximate expression for analyzing experimental data is readily apparent.

<u>Turbine-inlet temperature.</u> - If the efficiencies of the engine components are considered to be constant, the maximum thermal

efficiency of the engine is determined by the maximum allowable gas temperature at the entrance to the turbine; data should therefore be presented as a function of turbine-inlet temperature. A direct measurement of turbine-inlet temperature is both difficult and unsatisfactory. A reasonably accurate value, which serves as an index of the desired trends, can be obtained by making a direct measurement of the total temperature downstream of the turbine and adding the equivalent temperature drop of the compressor horsepower and the shaft horsepower.

$$T_3 = T_4 + \frac{c_{p,c}}{c_{p,t}} (T_2 - T_1) + \frac{550 \text{ shp}}{J \, c_{p,t} \, W_g} \tag{15}$$

ENGINE AND EXPERIMENTAL INSTALLATION

The engine used in these investigations was a J-33 turbojet engine, which consists of a double-entry centrifugal-flow compressor with an impeller tip diameter of 30 inches, 14 cylindrical-type combustion chambers, and a single-stage partial-reaction turbine with a nozzle-flow area of 0.84 square foot. The compressor rotor of the engine was assembled with a turbine-end drive shaft in place of the usual accessory-drive shaft. This alteration allowed one end of a high-speed inductor-brake shaft to be directly connected to the compressor rotor by the same type of floating coupling that is used between the compressor and the turbine. The standard accessory section, which contained starter, fuel pump, oil pump, and over-speed governor, was attached to the other end of the inductor-brake shaft. The air-flow characteristics of the compressor and the turbine were not altered. The engine was operated with the jet nozzle removed, making the tail-pipe-outlet area 22 percent larger than that of the standard engine, in order to increase the pressure drop across the turbine.

The engine was mounted on a bedplate suspended on cables and the thrust was measured by a strain-gage weighing device developed by the NACA. A sketch of the engine and the suspension arrangement is shown in figure 1.

Temperatures were measured at the compressor-inlet and compressor-outlet stations with iron-constantan thermocouples and at the turbine-outlet station with chromel-alumel thermocouples. Total and static pressures were measured at the compressor inlet, compressor outlet, burner outlet, and turbine outlet. Fuel flow was measured by a calibrated rotameter. Air flow was calculated from the turbine-outlet station measurements of total pressure, static pressure, and temperature.

EXPERIMENTAL PROCEDURE AND DATA CORRECTION

Runs were made by maintaining constant speed and increasing the fuel flow as the dynamometer load was increased. In order not to exceed the arbitrary operating limit for the turbine-inlet gas temperature of 2000° R, the corresponding turbine-outlet gas temperature calculated from equation (15) for the shaft horsepower absorbed by the dynamometer and estimated air flow was maintained.

All observed data were corrected to NACA standard sea-level conditions by use of the relations, corrected engine speed $N/\sqrt{\theta}$, corrected jet thrust F/δ, corrected shaft horsepower $shp/\delta\sqrt{\theta}$, corrected fuel flow $W_f/\delta\sqrt{\theta}$, corrected air flow $W_a\sqrt{\theta}/\delta$, corrected total temperature T/θ, and corrected total pressure P/δ, which may be derived by the methods given in reference 1.

RESULTS AND DISCUSSION

Basic data. - Shaft-horsepower data are presented for one engine tail-pipe configuration corresponding to a ratio of turbine-nozzle area to exhaust-nozzle area of 0.349; at this condition, the turbine-outlet static pressure was nearly atmospheric.

For purposes of comparison, data are also presented for the engine operating as a jet engine with a ratio of turbine-nozzle area to exhaust-nozzle area of 0.426 and a propeller-shaft power of zero.

Plots of the basic data of shaft horsepower and jet thrust per square foot of turbine-nozzle area as functions of turbine-inlet temperature for various compressor tip speeds and an area ratio of 0.349 are shown in figures 2 and 3, respectively. Compressor pressure ratio plotted against turbine-inlet temperature is shown in figure 4. The variation of fuel flow as a function of turbine-inlet temperature is presented in figure 5. In general, this relation was linear over the range investigated, which indicated that the burner efficiency was essentially constant. The variation of jet thrust of the engine with compressor tip speed at an area ratio of 0.426 and zero shaft horsepower is shown in figure 6. The particular engine used in these experiments had a basic corrected jet thrust of 4220 pounds per square foot of turbine-nozzle area at a corrected compressor tip speed U_c of 1510 feet per second.

Total equivalent horsepower. - The shaft horsepower, jet thrust, and total equivalent horsepower are presented as functions of compressor pressure ratio in figure 7 for a constant turbine-inlet temperature of 2000° R. The curves in figure 7 have been generalized on the basis of 1 square foot of turbine-nozzle-outlet flow area. The jet thrust increases with increasing compressor pressure ratio to a maximum of 3450 pounds per square foot of turbine-nozzle-outlet flow area at a compressor pressure ratio of 3.54. The shaft horsepower reached a maximum of 1650 at a compressor pressure ratio of 3.2 (compressor tip speed, 1293 ft/sec) and then decreased slightly to 1600 horsepower at a pressure ratio of 3.54 (compressor tip speed, 1438 ft/sec). This effect is probably due to the establishment of sonic flow at some point in the turbine.

Thermal efficiency. - The ability of the engine to utilize the heat energy of the fuel is revealed by an inspection of the thermal efficiency developed. A plot of the variation of thermal efficiency of the engine with variable turbine-inlet temperature is shown in figure 8. The thermal efficiency is calculated from the shaft horsepower, the jet thrust, and the fuel flow by means of equation (11). The increase in thermal efficiency that could be obtained by having a turbine-inlet temperature limit higher than 2000° R is indicated by the slopes of the curves.

Ratio of shaft horsepower to total available power. - The variation of ratio of shaft horsepower to total available power φ in an engine having a fixed relation between compressor tip speed and turbine tip speed is determined by the equilibrium running condition of all the engine components and cannot be reliably determined except from experimental data. The variation of measured values of φ is shown as a function of turbine-inlet temperature and compressor tip speed in figure 9. A cross plot that shows the variation of φ with compressor tip speed for an area ratio of 0.349 with a turbine-inlet temperature of 2000° R is presented in figure 10. The range of φ varies from 0.512 at a compressor tip speed of 1023 feet per second to 0.313 at a compressor tip speed of 1438 feet per second. The trend of the curve in figure 10, as well as the apparent maximum shaft horsepower shown on figure 7, is the result of a variation in the capacity of the turbine to extract power as well as a possible change in adiabatic efficiency.

Static thrust with propeller. - In order to estimate the take-off performance of a turbine-propeller engine, the propeller-shaft horsepower must be evaluated in terms of thrust produced. A study of static-thrust-calibration data for propellers of modern high-performance design reveals that the pounds of static thrust produced is approximately equal to four times the horsepower input. A plot of total equivalent thrust as a function of compressor tip speed is shown in figure 11. The curve of jet thrust against compressor tip speed for the 0.426 turbine-area ratio and zero shaft horsepower is included in order to indicate the gain in static thrust that could be achieved by use of a propeller on this particular engine. The static thrust could be 9150 pounds per square foot of turbine-nozzle area at a compressor tip speed of 1438 feet per second or better than two and one-half times the thrust of the turbojet engine alone. The variation of thrust specific fuel consumption with compressor tip speed based on jet thrust plus propeller thrust is shown in figure 12. At a compressor tip speed of 1438 feet per second, a value of φ of only 0.312 (fig. 10) has reduced the thrust specific fuel consumption from 1.27 pounds fuel per pound thrust for the straight jet engine to 0.58 pound fuel per pound thrust for operation as a turbine-propeller engine.

SUMMARY OF RESULTS

A compressor-burner-turbine combination was operated over a range of turbine-inlet temperatures up to a maximum of 2000° R, compressor tip speeds up to a maximum of 1438 feet per second and with a tail-pipe-outlet area 22 percent larger than that of the conventional J33 engine. The shaft horsepower was a maximum of 1650 horsepower per square foot of turbine-nozzle area at a compressor pressure ratio of 3.2 (compressor tip speed, 1293 ft/sec) and turbine-inlet temperature of 2000° R, and then declined slightly to 1600 horsepower per square foot of turbine-nozzle area at a compressor pressure ratio of 3.54 (compressor tip speed, 1438 ft/sec) and turbine-inlet temperature of 2000° R. As the compressor tip speed was increased from 1023 to 1438 feet per second at a constant turbine-inlet temperature of 2000° R, the ratio of shaft work to total useful work dropped from 0.512 to 0.313.

An empirical propeller relation that thrust is equal to four times the horsepower input indicates that, at a compressor tip speed of 1438 feet per second, a static thrust of 9150 pounds per square foot of turbine-nozzle area, or more than two and one-half times the static thrust of the conventional turbojet engine, is available from the engine investigated. At a compressor tip speed of 1438 feet per second, specific fuel consumption decreased from 1.27 pounds fuel per pound thrust for operation as a conventional jet engine to 0.58 pound fuel per pound thrust for operation as a turbine-propeller engine.

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REFERENCE

1. Sanders, Newell D.: Performance Parameters for Jet-Propulsion Engines. NACA TN No. 1106, 1946.

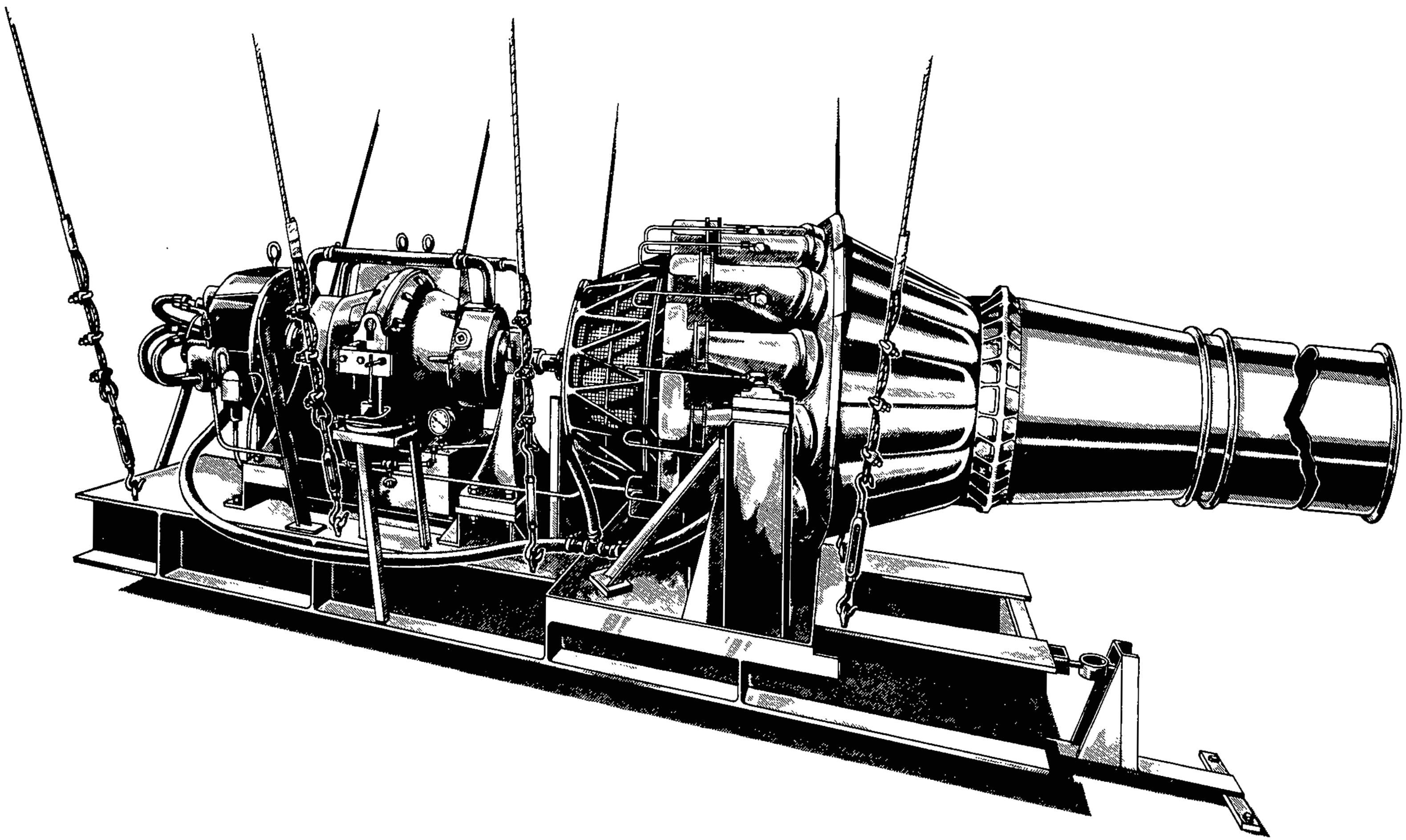

Figure 1. - Turbine-propeller engine and suspension arrangement.

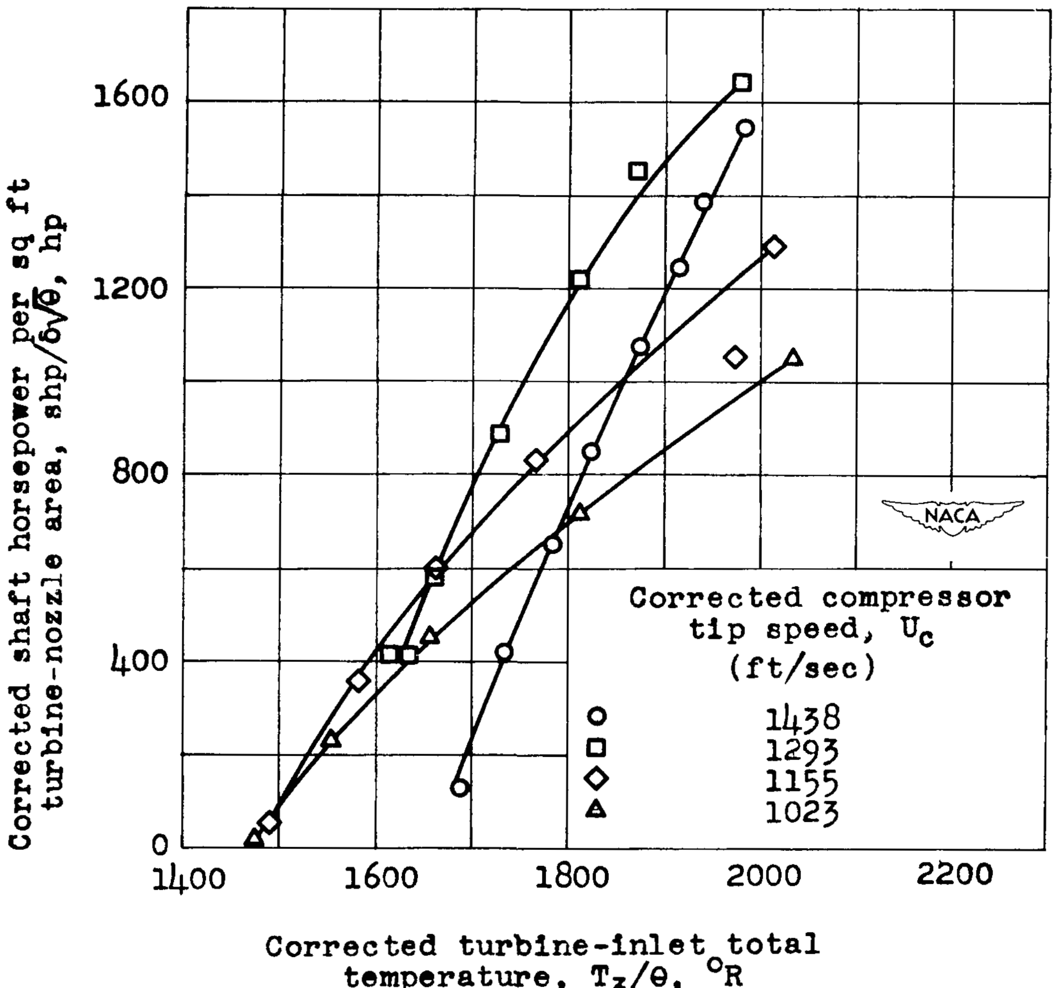

Figure 2. - Variation of corrected shaft horsepower per square foot of turbine-nozzle area with corrected turbine-inlet total temperature for ratio of turbine-nozzle area to exhaust-nozzle area of 0.349 and various corrected compressor tip speeds.

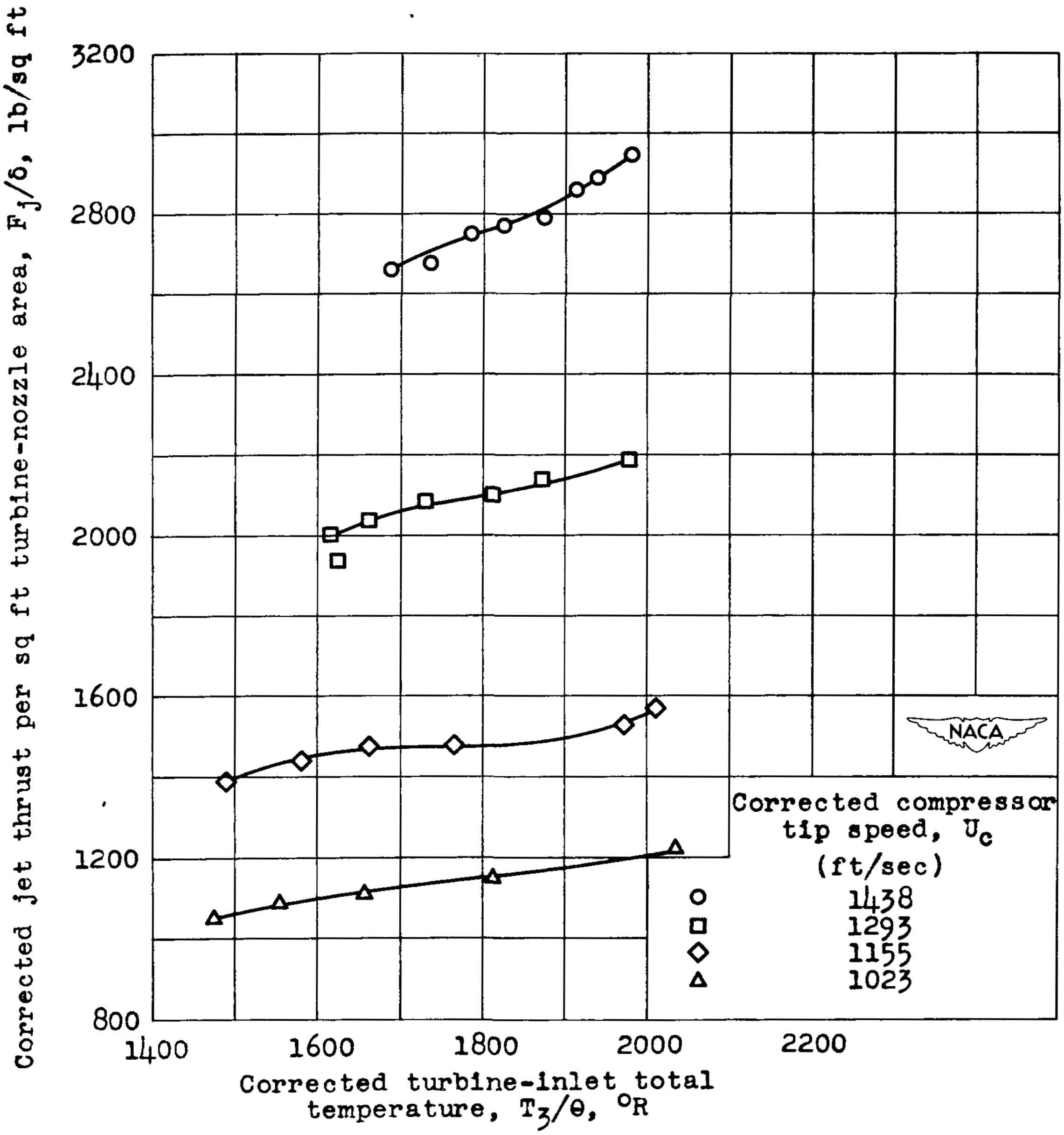

Figure 3. - Variation of corrected jet thrust per square foot of turbine-nozzle area with corrected turbine-inlet total temperature for ratio of turbine-nozzle area to exhaust-nozzle area of 0.349 and various corrected compressor tip speeds.

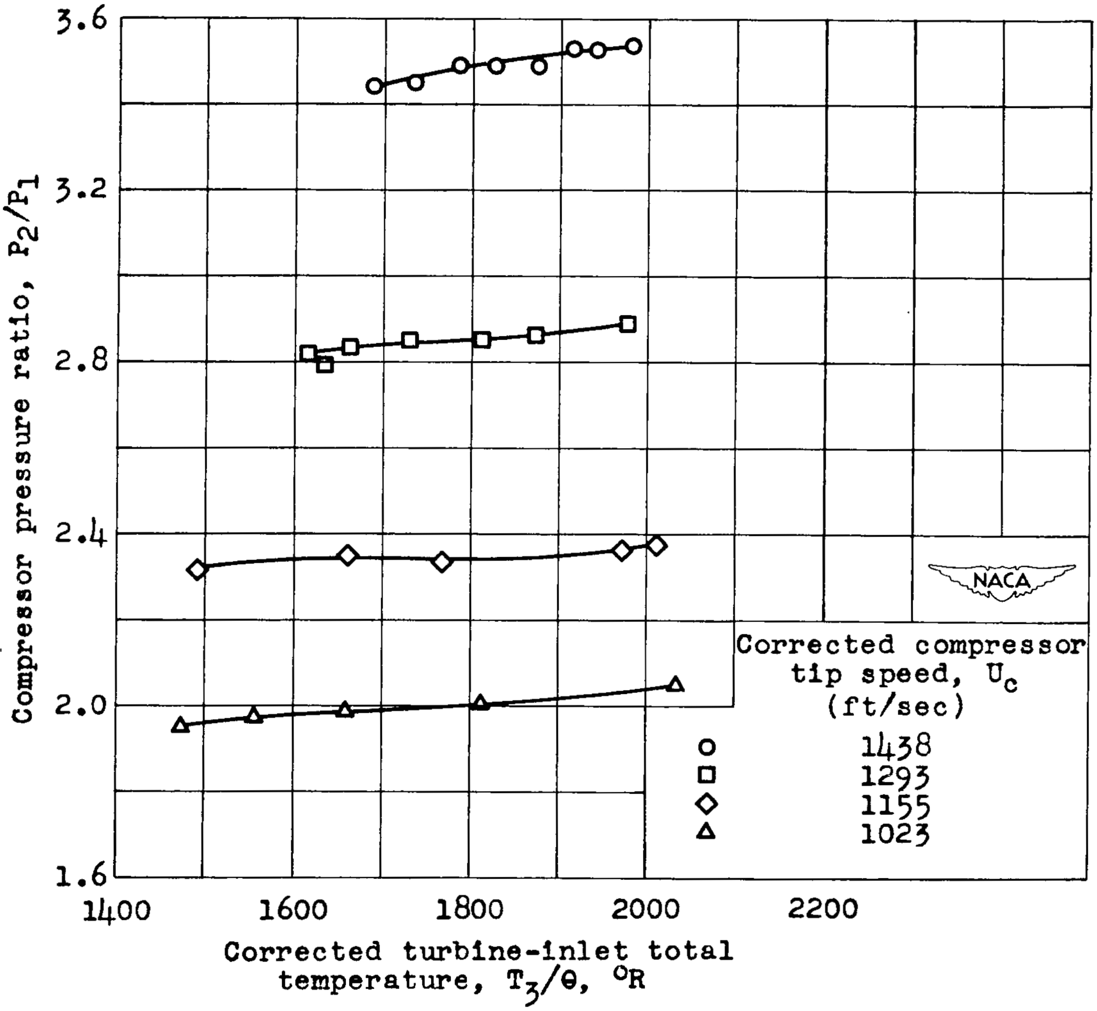

Figure 4. - Variation of compressor pressure ratio with corrected turbine-inlet total temperature for ratio of turbine-nozzle area to exhaust-nozzle area of 0.349 and various corrected compressor tip speeds.

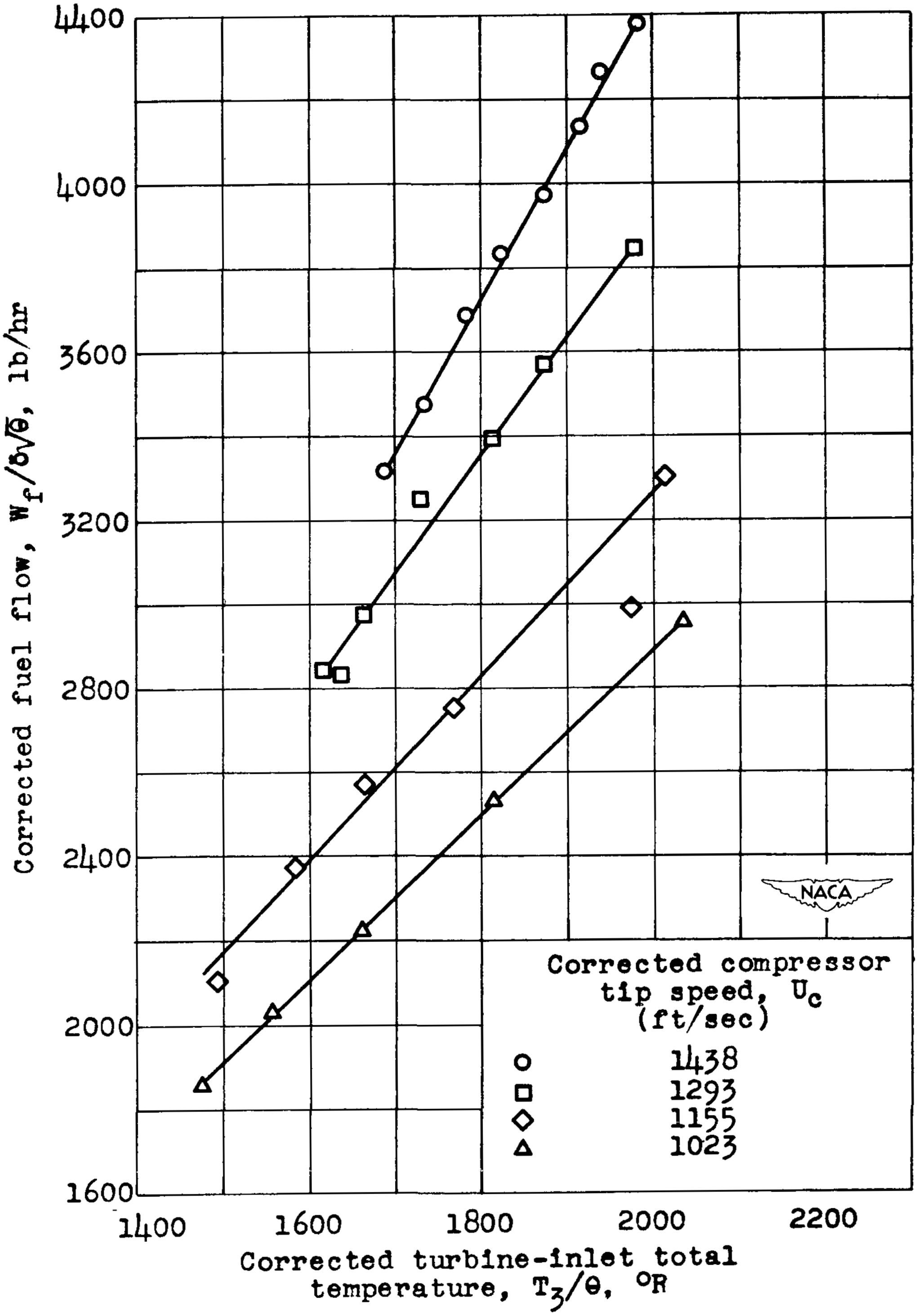

Figure 5. - Variation of corrected fuel flow with corrected turbine-inlet total temperature for ratio of turbine-nozzle area to exhaust-nozzle area of 0.349 and various corrected compressor tip speeds.

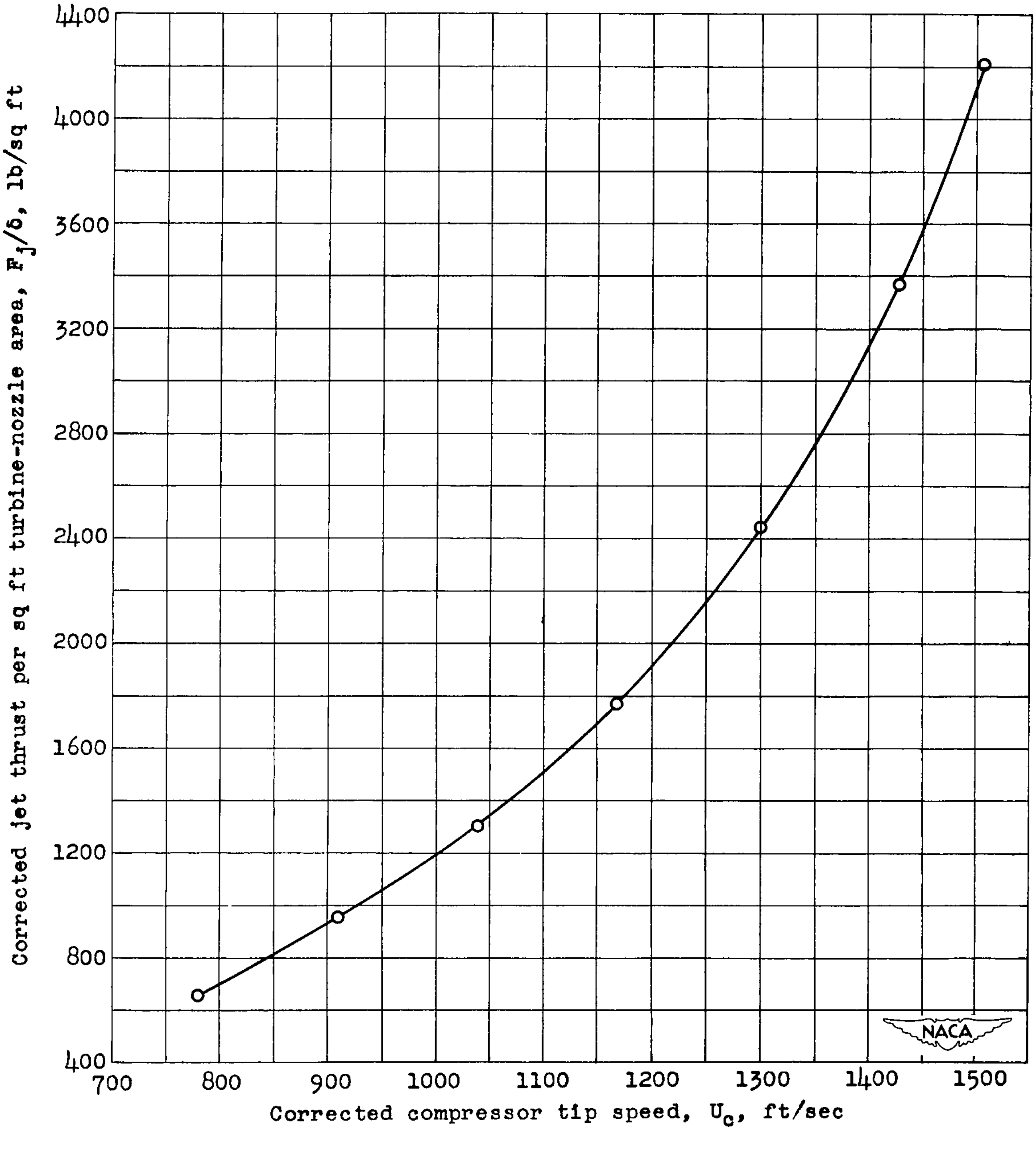

Figure 6. - Variation of corrected jet thrust per square foot of turbine-nozzle area with corrected compressor tip speed for ratio of turbine-nozzle area to exhaust-nozzle area of 0.426 and zero shaft horsepower.

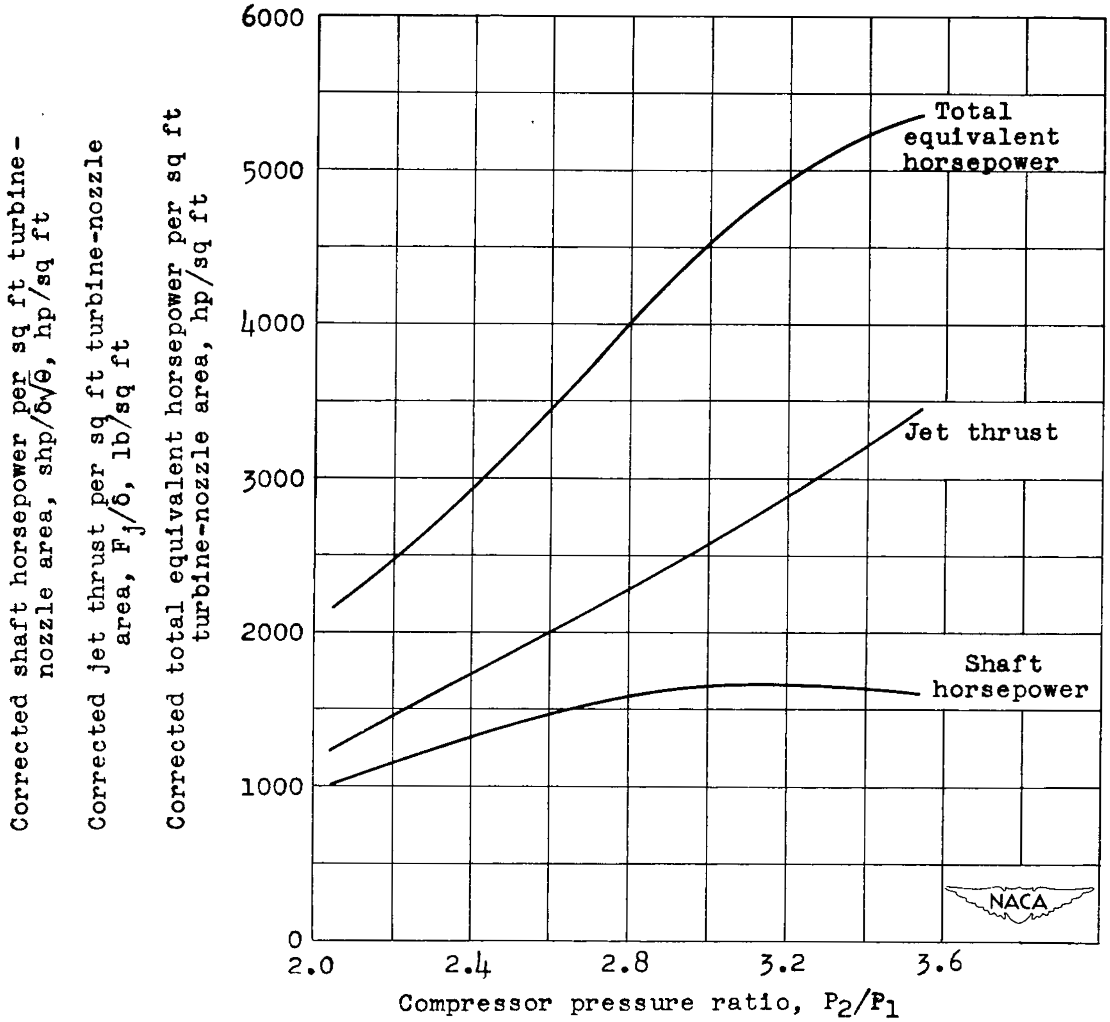

Figure 7. - Variation of horsepower and thrust per square foot of turbine-nozzle area with compressor pressure ratio for corrected turbine-inlet total temperature of 2000° R and ratio of turbine-nozzle area to exhaust-nozzle area of 0.349.

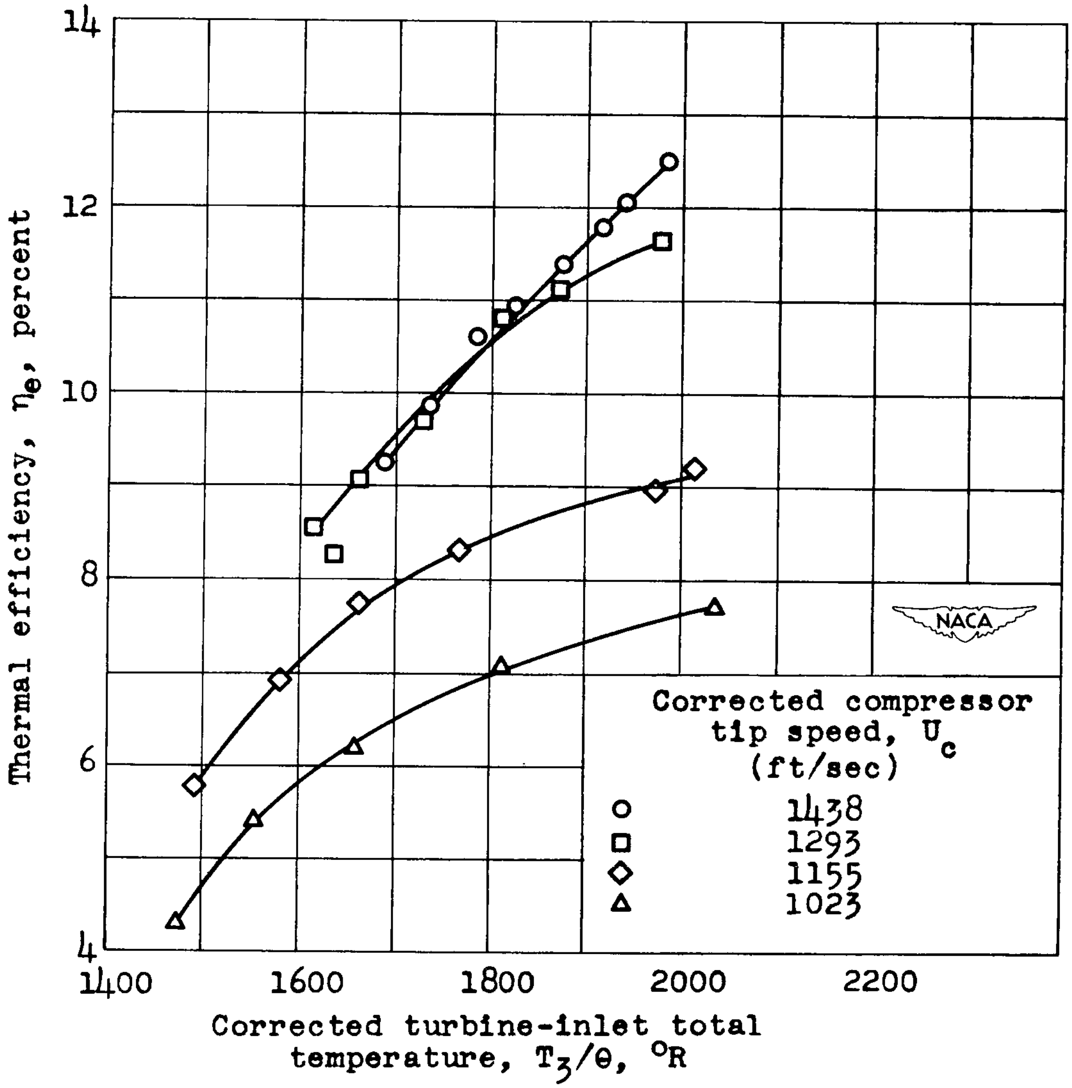

Figure 8. - Variation of thermal efficiency with corrected turbine-inlet total temperature for ratio of turbine-nozzle area to exhaust-nozzle area of 0.349 and various corrected compressor tip speeds.

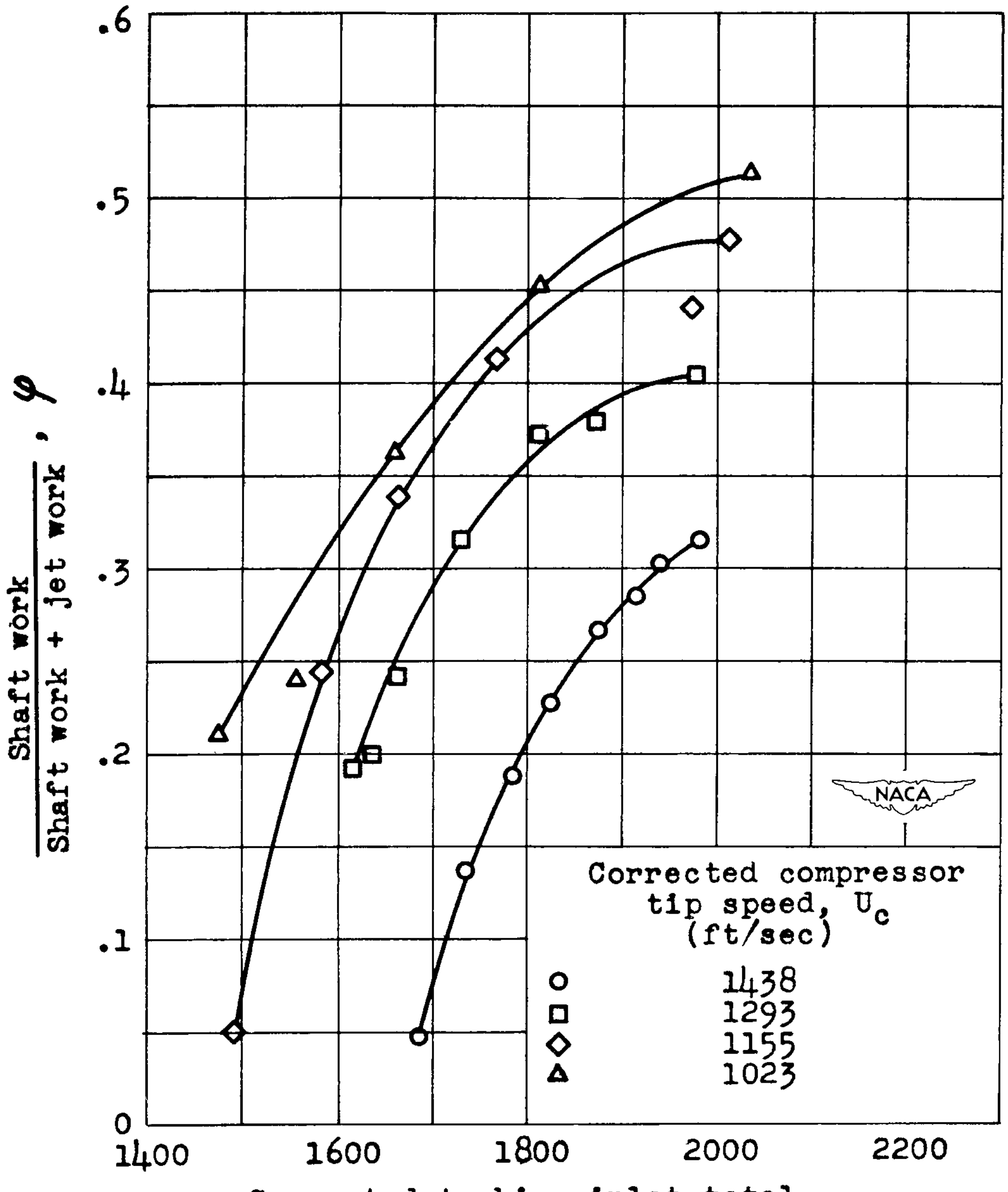

Figure 9. - Variation of $\frac{\text{shaft work}}{\text{shaft work + jet work}}$ with corrected turbine-inlet total temperature for ratio of turbine-nozzle area to exhaust-nozzle area of 0.349 and various corrected compressor tip speeds.

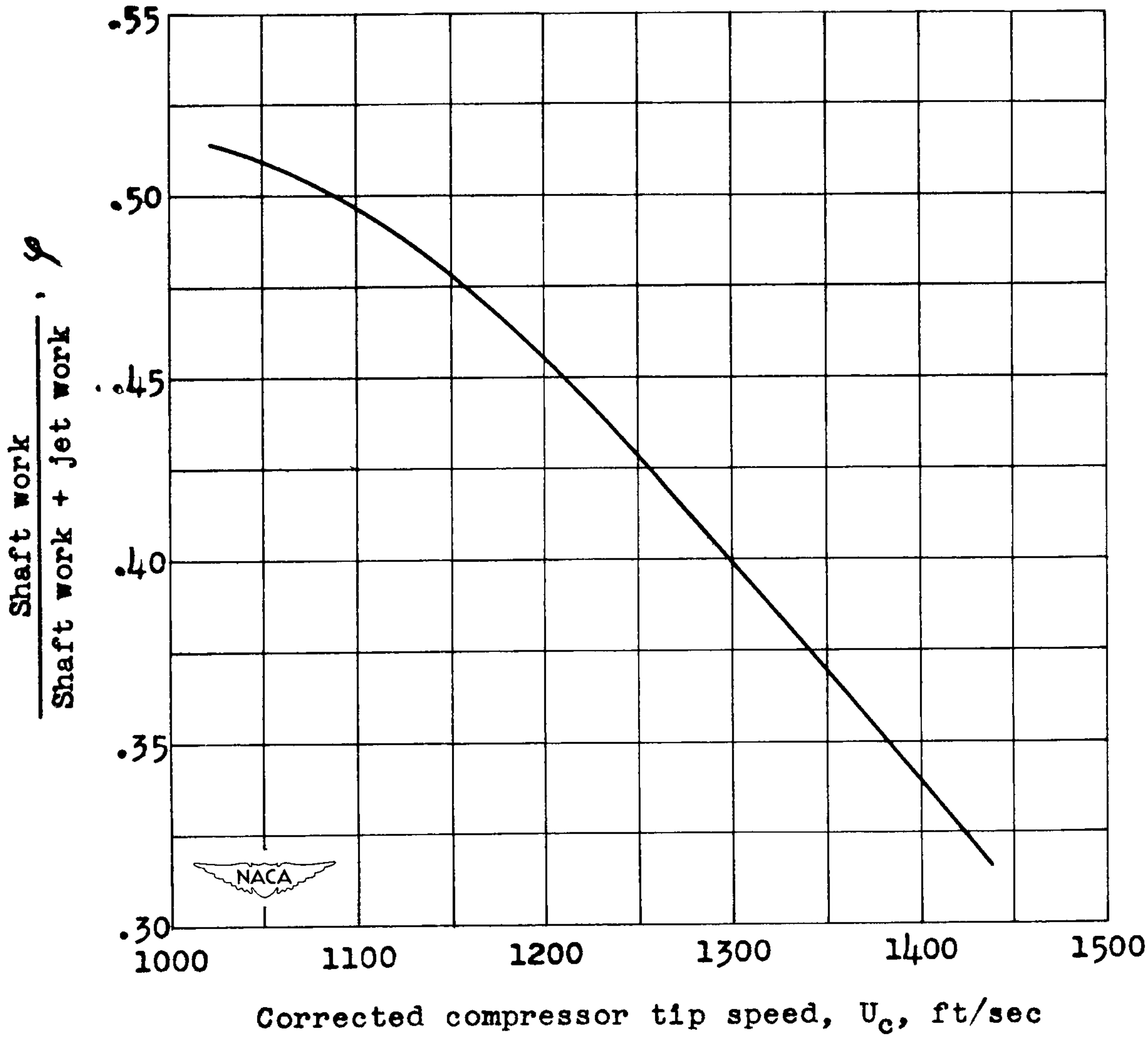

Figure 10. - Variation of $\frac{\text{shaft work}}{\text{shaft work + jet work}}$ with corrected compressor tip speed at corrected turbine-inlet total temperature of 2000° R with ratio of turbine-nozzle area to exhaust-nozzle area of 0.349

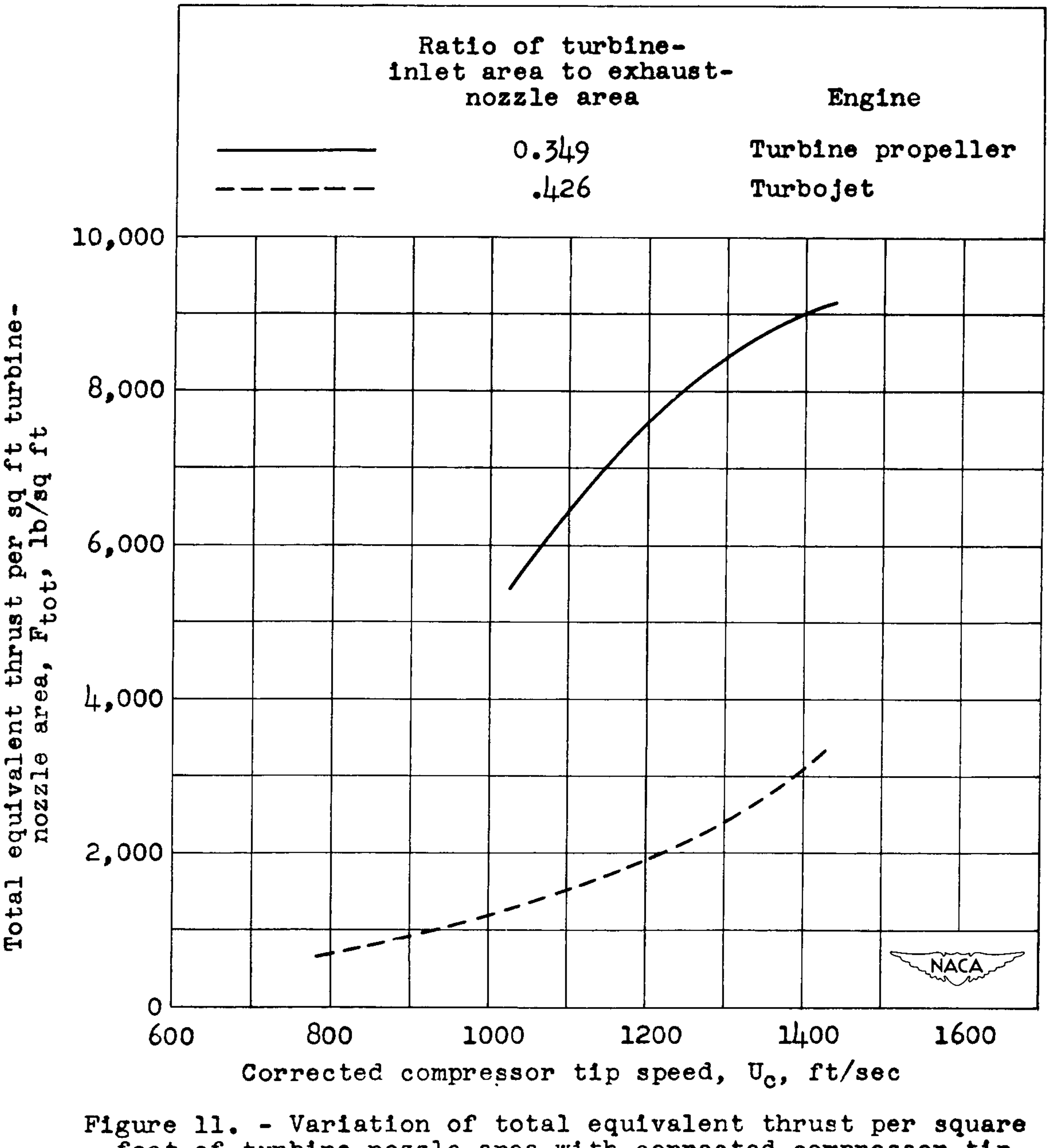

Figure 11. - Variation of total equivalent thrust per square foot of turbine-nozzle area with corrected compressor tip speed for corrected turbine-inlet total temperature of 2000° R.

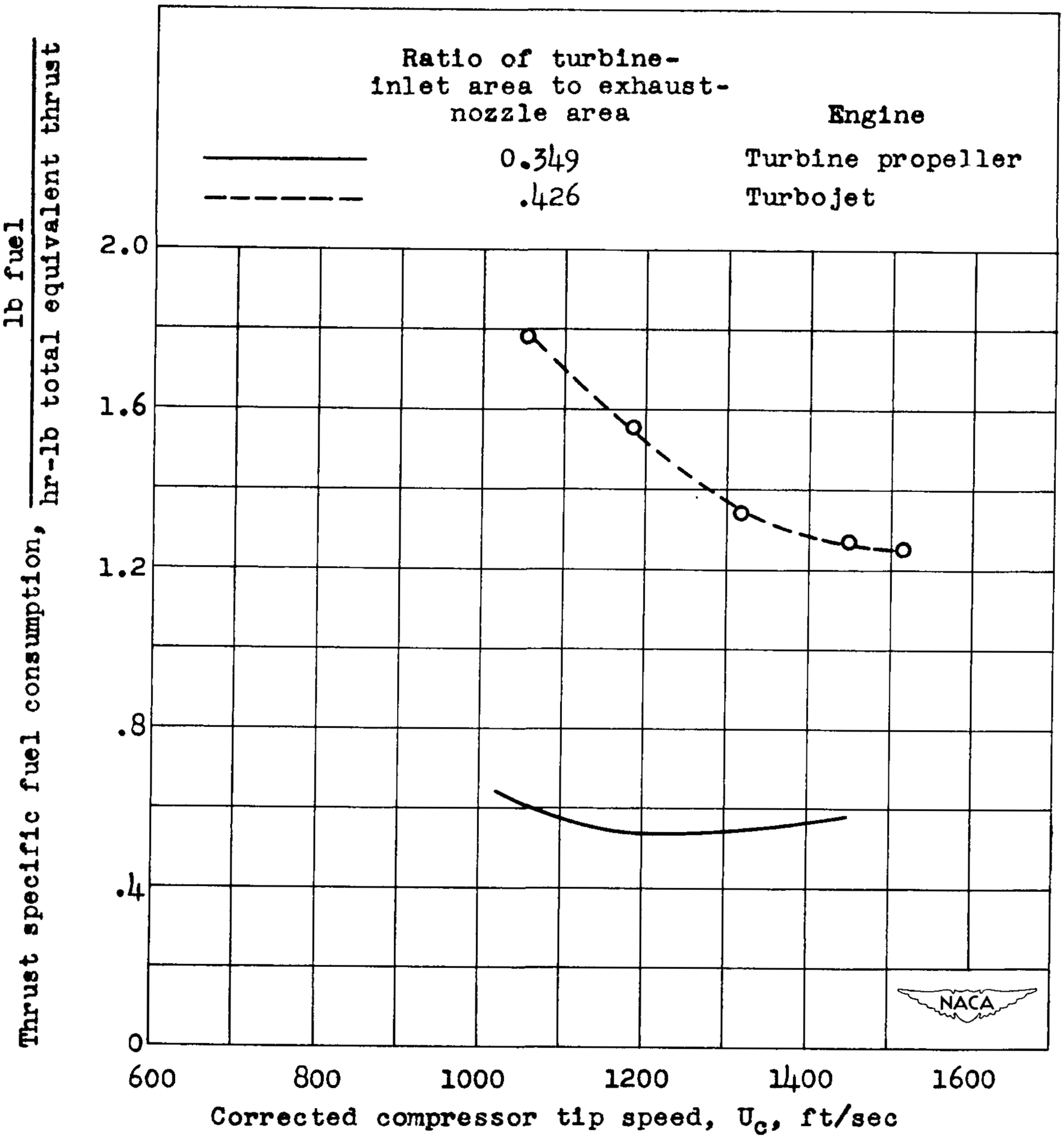

Figure 12. - Variation of thrust specific fuel consumption with corrected compressor tip speed for corrected turbine-inlet total temperature of 2000° R and two turbine-nozzle area to exhaust-nozzle area ratios.